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MARS TO ORBIT WITH PUMPED HYDRAZINE

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Abstract

A propulsion point design is presented for lifting geological samples from Mars. Vehicle complexity is kept low by choosing a monopropellant single stage. Little new development is needed, as miniature pump fed hydrazine has been demonstrated. Loading the propellant just prior to operation avoids structural, thermal, and safety constraints otherwise imposed by earlier mission phases. Hardware mass and engineering effort are thereby diminished. The Mars liftoff mass is 7/8 hydrazine, <5% propulsion hardware, and >3% each for the payload and guidance.

Introduction

Humanity looks forward to Mars Sample Return missions, currently scheduled to occur within the next decade. However, the rocket problem of ascending from Mars has remained unsolved, in the absence of a dedicated long term effort. The applicability of advances already tested for similar maneuvering needs is thus a primary consideration.

Recent work on Mars ascent includes a requirements analysis with comparisons to the limits of conventional propulsion.[1] Space-qualified hardware offers no low-risk options, as it does for planetary flybys, orbiter missions, and Mars landing. This fact is independent of the stage count. Producing propellants on Mars has been widely discussed, but addresses a different problem. The critical unknown is how to build miniature rocket stages having very high propellant fractions and enough thrust.

Mars departure is more than twice as hard as leaving earth's moon, with respect to the two fundamental maneuvering parameters (Δv and acceleration). Earth launch vehicles and their stages stand alone in having the necessary capability. Of course they are many times too heavy to be affordably placed on a Martian launch pad. Therefore, Reference 1 discussed the application of launch vehicle design principles on a tiny scale. A hypergolic bipropellant engine fed by piston pumps was schematized. Other possible solutions under consideration also require new advances. Pressure fed bipropellants need engines, structure, and tanks to all be lighter than the state of the art.[2] A combination of ideas from References 1 and 2 would simplify pump fed engine development.[3] Even so, a high pressure long life bipropellant thruster is not available. Currently, NASA's reference design for a Mars Ascent Vehicle (MAV) uses a multi-stage solid rocket, which imposes its own unique set of challenges.[4]

This paper offers another in the series of possibilities. Despite its low specific impulse (Isp), hydrazine can propel a MAV to orbit with a small payload. Advantages of this monopropellant choice include the availability of a lightweight thruster, the simplicity of a single-tank configuration, and the ease of fueling on Mars. Moreover, miniature pumped hydrazine systems have already been tested and flown experimentally.[5-7] This experience lends realism to the MAV propulsion design presented herein.

MR-125 Thruster

Beginning in 1988, a high pressure hydrazine thruster was developed by Primex Aerospace Company (then Rocket Research) for LLNL and used on advanced technology programs. The MR-125 (MR = Monopropellant Rocket) is shown in Figure 1, with data in Table 1.

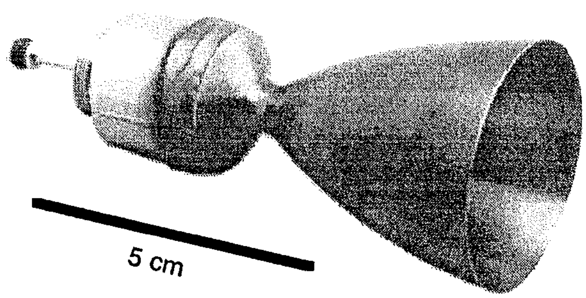

Figure 1. The Primex MR-125 Hydrazine Thruster.

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Table 1. MR-125 Test Parameters.

Equilibrium Vacuum Performance	
Thrust	250 N (56 lbf)
Specific Impulse	240-245 s
Feed Pressure	7.2 MPa (1050 psia)
Chamber Pressure	4.1 MPa (600 psia)
Chamber Temperature	1380 K (2025 F)
Response Times After Warmup	
Start Response	<15 msec to 90% thrust
Stop Decay	<20 msec to 10% thrust
Pulse Repeatability	±6% for 20 msec pulses
Shortest Pulses Tested	5 msec, ±13% repeatability
Off Pulsewidth Capability	5-10 msec for flight control
Life Tested Thruster	
Total Propellant Throughput	93 kg (205 lbm)
Total On Time	994 s
Longest Single Burn	600 s at nominal conditions
Ambient Temperature Starts	3
Feed Pressure Range	130-1400 psia
Design Parameters	
Mass (as pictured in Fig 1)	100 grams
Expansion Ratio	50:1
Hydrazine Grade	Purified

It is by far lighter and smaller than conventional thrusters, and Isp is improved. These advantages are directly attributable to a high operating pressure, which reduces nozzle area proportionately. Also, the density of the decomposition gases is greater than in conventional low pressure thrusters, which results in a longer residence time for a given catalyst bed geometry. Weight is additionally reduced by the omission of features included on satellite thrusters. For example, the latter need heaters and thermal shielding to maintain pulse performance over many years.

Over thirty MR-125 units were manufactured for use in experimental propulsion system tests including pump-fed suborbital flight. Lightweight valves were developed by Moog Inc. for use with this high pressure thruster. A cold gas piloted design and a warm gas piloted model massed under 40 grams each. The combined delay of both valve stages was between 2 and 5 msec. These custom valves were used in system tests, and timing data in Table 1 reflect this. The weights of the valve, thruster, and thermal standoff sum to just over a half percent of thrust.

The MR-125 thruster and a similar 500-N (112 lbf) version (MR-133) have been considered for various NASA Mars landing missions (1998-2003). The demonstrated weight savings and performance would be of great benefit to the descent propulsion systems. However, the high operating pressure has been a challenge for mission designers due to increased tank weight and the limited availability of other high pressure components.

Like the MR-125, the MR-133 has completed development including thrust stand characterization and life testing. Flight qualification for either would include random vibration, pyroshock, and thermal mapping. From a structural and thermal design standpoint, the qualification environments are not expected to pose any program risk.

Given the benign reaction temperature of monopropellant, material thermal limits don't impair lifetime as with miniature high pressure bipropellant thrusters. As shown in the table, one of the hydrazine thrusters operated several times longer than needed for Mars ascent.[8] In addition, pulse performance is adequate to effect directional control during the flight. Short off pulses would always be crisp, since the catalyst would remain hot.

In order to capitalize on the attributes of the MR-125 thruster, it will be beneficial to apply all possible means to reduce the mass of its feed system and other MAV components. Several distinct but synergistic ways for doing this are treated in the rest of this paper.

Fueling on Mars

Sending a fully fueled MAV from earth to Mars would be in keeping with conventional spacecraft methodology, but doing so imposes significant constraints beyond those of the Mars ascent itself. Table 2 lists the earlier mission phases chronologically, along with the design impacts of carrying the propellant from earth in the MAV.

Personnel safety during prelaunch operations is of primary importance. It must be established, to a high statistical degree of certainty, that a spacecraft on earth could not possibly expel fluids by leakage or rupture. Given the toxicity of commonly used propellants, ordinary pressure proof tests and leak tests alone are inadequate. Rigorous fracture mechanics requirements limit tank wall thinning.

It is similarly necessary to guarantee that thrusters cannot operate unexpectedly. Valves of different types must be placed in series and controlled by different circuits to inhibit propellant flow. This multiple fault tolerance adds significant mass to both mechanical and electrical systems.

A MAV having pre-filled tanks must withstand high structural loads due to the multiplication of liquid mass by severe acceleration environments. Propellant storage for many months requires extremely low leakage rates.

Satellites routinely carry all the above capabilities. The extra features are not usually viewed as heavy, but for only one reason. All spacecraft maneuvers performed to date in earth orbit and beyond are much easier than Mars ascent.

Table 2. Heavy items to be avoided by fueling on Mars.

Mission Phase:	Full Tank Requires:
Earth prelaunch	Thicker tank wall to meet fracture mechanics rules. Multiple valves in series with independent controls. Pressure monitoring.
Earth launch	Tank and structure to withstand vibration.
Trans-Mars cruise	Nine month leaktight propellant storage.
Mars entry & descent	Loaded tank designed for >30 m/s^2 deceleration.
Mars surface stay	Tank heaters and insulation to prevent propellant freezing.
Mars prelaunch	Startup pressurization on MAV if no fluid connections to lander.

Each item in Table 2 would result in additional hardware on a pre-fueled MAV, i.e. less rock and soil, quite possibly negative quantities. Cost and schedule would be impacted as well. Therefore, the MAV described herein remains empty until just before Mars departure. A MAV designed to perform only its mission role can be very lightweight.

Fueling on Mars is particularly synergistic with a pumped hydrazine MAV. Only one fluid needs to be transferred, at low pressure. The propellant is already on most landers (Viking, Mars Polar Lander, Mars 2001 and 2003). The landing tanks might be enlarged, or storage capacity would be included as launch support equipment.

While this approach may burden lander design, the rocket equation favors it from a mission mass standpoint. Consider a hypothetical combination of a lander and a prefueled MAV. Next consider changes with the rock sample size and total mission mass held fixed. Each kilogram of hardware that can be shaved off the MAV increases the support equipment mass allowance by 7 kg. The leverage comes from avoiding 6 kg of hydrazine needed to lift the nonessential kilogram to Mars orbit.

The simplistic break even point occurs when a Mars-filled MAV tank can be one seventh lighter than an earth-filled tank. This minor improvement allows for ascent propellant tanks on the lander. The reality is far better, yielding a net mission benefit by fueling on Mars.

The above calculations assume a single hydrazine stage, but the mission mass advantage of fueling on Mars is of a similar character in other cases. A broader benefit of performing automatic fluid transfer is that the experience will be relevant to in-situ propellant production (ISPP).

Single Stage Advantages

From a vehicle engineering perspective, a single stage MAV is attractive for several reasons. It is cumbersome to stack stages vertically within the Mars arrival capsule. A short MAV fits better. Second, there is less hardare to design, build, and test. Third, loading fuel and pressurant automatically on Mars then subsequently disconnecting is more complicated for an upper stage.

Any upper stage would require additional miniaturization, along with extra connecting structure and separation mechanisms. Obviously, increasing the stage count does relax the propellant fraction requirement. Reference 1 quantified this trade, and found no strong preference for staging over the avoidance of further miniaturization.

From a mission perspective, there is yet a fifth reason to consider only one stage. The sample's orbital parameters must be determined to permit retrieval and transfer to earth. To put it simply, larger objects are easier to find. For a given size MAV, it may be desirable to leave the entire vehicle attached to the precious sample until it is located.

Mars and Earth SSTO Compared

Despite the above logic, it is acknowledged that there is a natural negative reaction to the suggestion of using only

Table 3. Earth SSTO Compared to Single Stage MAV.

	Earth SSTO, LOX-hydrogen	Mars SSTO, hydrazine
Maneuvering needs:		
Vacuum Equivalent Δv to Low Orbit	10,000 m/s (large vol/surf)	4500 m/s (small vol/surf)
Thrust ÷ Launch Mass	>13 m/s^s	10 m/s^2
Propulsion Parameters:		
Vacuum Isp	450 s	235 s
Propellant Fraction	**0.896**	**0.858**
Propellant Density	0.36	1.00
Best Ratio of Propellant Mass to Tank Mass	47 (Al-Li cylinder, shuttle actual)	220 (Ti sphere, 75 ksi at 50 psi)
Thrust/Weight of Pump Fed Engines (vac, earth)	70 (Vulcain, SSME, etc.)	50 to 70 range (experimental)
Design Features:		
Tank Configuration	odd shapes	one sphere
Non-Tank Structure	extensive	minimal
Recovery Items Carried	wings, TPS, wheels, etc.	none
Mass Impact of Safety	significant	negligible
Mass for Reusability	measureable	none
Scale of Implementation	1000 tons	<<1 ton

one stage at low Isp to perform a large velocity change. Therefore, it is interesting to compare single stage Mars ascent with earth SSTO (single stage to orbit). Table 3 indicates that Mars SSTO should be easy by comparison.

Velocities for Mars escape and orbiting are approximately half those for earth. No credit is taken in Table 3 for Mars' thinner atmosphere, or for relatively lower gravity losses due to doubling the ratio of thrust to local weight on Mars. It is effectively assumed that drag losses on a tiny vehicle would be comparable to earth drag losses on a large vehicle, consistent with physical scaling.

The net result of roughly halving both Δv and Isp is that the Mars SSTO vehicle needs a propellant fraction of 86%, slightly lower than its earth counterpart. Hydrazine's nearly threefold higher density than the cryogenic oxygen-hydrogen combination contributes strongly to reducing relative tank mass. The strength/weight ratio of titanium combined with a more efficient spherical shape can lighten the MAV tank even further as indicated.

The last section of Table 3 is qualitative, but the differences suggest that it would be easier still for the single stage MAV to meet its hardware mass budget. Only the last item raises uncertainty, so details are explained below.

MAV Sizing and Operation

A 200 kg Mars liftoff mass is arbitrarily chosen. It may be scaled somewhat to fit a particular Mars Sample Return mission. In order to meet thrust-to-weight and velocity change requirements, the propulsion system delivers 2000 N thrust in vacuum, and carries 175 kg of hydrazine. Table 4 lists a mass summary along with calculated capability. A mass margin and residual fluids are included.

Feeding MR-125 thrusters with a small high performance pump is the key to holding mechanical hardware within the allotted 8 kg. As on launch vehicles, low pressure tanks and high pressure engines are lighter than pressure fed alternatives, by many times the pump mass.

Figure 2 shows a fluid schematic. The system is based on the results of previous pump-fed hydrazine tests.[5-7] Pump chambers are alternately filled at tank pressure, and expelled at a much higher pressure. Gaseous decomposed hydrazine powers the pump then is exhausted externally. The operating principle is a gas generator cycle, per the terminology used to classify launch vehicle engines. The net Isp is 2 percent below that of the thrust chambers, since this fraction of the expended mass powers the pumps. A still smaller fraction of the high pressure warm gas is regulated down for in-flight tank pressurization.

Table 4. Mass and Performance Summary.

Propulsion Hardware & Structure		8 kg
Batteries & Valve Electronics		1
Guidance & Control		6
Sample Package		6
Mass Margin		4
Total Dry Mass		25
Hydrazine		175
Total Mars Liftoff Mass		200 kg
System Isp	Residual Fluids	Δv
235 s	2 kg	4612 m/s
230	2	4514
235	3	4528

A preliminary system analysis yielded the operating parameters displayed in Table 5. Ideal gas calculations were used, because liquid displacement occurs at steady pressures without gas expansion work. Pressure drops through the gas generator circuit require liquid to be boosted to a higher pressure than the gas. This can be accomplished by using differential area pistons running in pump gas cylinders which are larger than the liquid cylinders. It is assumed here that the piston area ratio is 1.56, as successfully tested previously. The gas volume required to run the pump is additionally 20% higher due to flow losses during gas and liquid valve switching.

Extra valves normally included on spacecraft are absent from the schematic. A tank isolation valve would be a particular burden, since the low pressure pump feed tube must be large. Conventional fill valves having redundant sealing features are replaced by a flyaway disconnect. This has a valve which closes itself at launch.

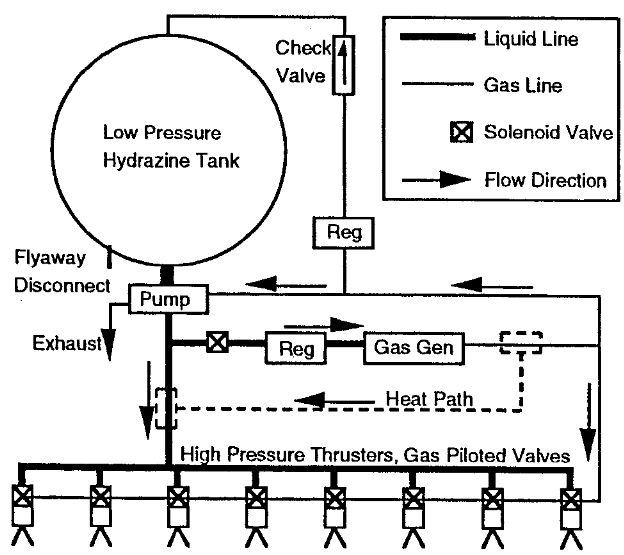


Figure 2. Pump Fed Hydrazine Propulsion Schematic.

Table 5. Propulsion Operating Parameters.

Propellant Allocations and Mass Flows			
Main Thrusters	169.3 kg	846 g/s	(2000 N at Isp=241 s)
Pump Power	3.7	19	(1600 cc/s volume flow)
Tank Pressurant	0.3	1.5	(867 cc/s into tank)
Unused Liquid	1.7		

Total burn time at full thrust: 173 kg + 0.865 kg/s = 200 s
Net Isp = 241 x (846)/(846 + 19) = 235 s (no pump exhaust thrust)

Pressures, Temperatures, and Densities				
Pump Liquid Cyl	7.6 MPa	300 K	1001 kg/m^3	
Thruster Feed	7.2	306	995	
Thrust Chamber	4.1	1380	5.4	(m.w. = 15)
Gas Gen Output	5.7	1000	8.2	(m.w. = 12)
Pump Gas Cyl	5.0	600	12.0	(m.w. = 12)
Tank Ullage	0.35	300	1.7	(m.w. = 12)

Heat Rates	
Gas to Liquid in Heat Exchanger	20 kW
Pressurant to Tank Wall and Propellant	1 kW

Volume Displacements

Tank, 175 kg hydrazine at >990 kg/m^3 requires 177 liters + ullage
Pump, 880 cc/s displaced by 4 cylinders x 22 cc at 10 Hz
Mission life of the pump is 10 Hz x 200 s = 2000 cycles

Once the MAV tank and feedlines contain propellant, only the small solenoid valve in the gas generator circuit needs to be actuated to start the system. Initially, warm gas at tank pressure powers the pump. This amplifies the pump discharge pressure in a positive feedback loop which rapidly raises the system to operating pressure as controlled by the liquid regulator.

Before liftoff, all catalyst beds are warmed up by sending short electrical pulses individually to the thrust chamber valves. The warmup propellant consumed need not detract from that available for ascent. Instead, the lander continues to top off the MAV tank until after all thrusters are hot.

The gas is cooled substantially in a heat exchanger, to reduce the operating temperature of the pump seals. This cooling strategy is reflected in the table and in the schematic. Note that the gas generator temperature is already far below that of the thrust chambers, by virtue of a high fractional ammonia dissociation. However, 1000 K is still too hot for a long seal life with negligible leakage. As indicated, the warm gas would be passed through a heat exchanger then on to the pump at 600 K. The resulting warming of the propellant by 6 K would enhance thruster performance.

During flight, only 300 grams of gas is needed to pressurize the tank. The actual quantity may be less if it remains above 300 K. This is in stark contrast to "warm gas tank pressurization" in the pressure-fed sense. Kilograms of decomposed hydrazine would accumulate in the tank if it fed the MR-125 thrusters directly. Reduced pressurant is another mass advantage of pump fed engines.

The heat conveyed to the low pressure tank is also comfortably low. This is important due to concerns of hydrazine thermal decomposition above 480 K. Even if the tank wall had to radiate the entire kilowatt listed in Table 5, it would remain below 350 K at high emissivity.

Stage Layout and Component Designs

The single stage propulsion configuration is depicted in Figure 3. It comprises one spherical tank, eight high pressure thrusters, and one pump. Components are mounted below the tank for several reasons. Obviously, the pump should be at the tank outlet port. The thrusters are clustered closely around the pole to minimize resulting vehicle torques. Plumbing weight is minimized by locating all other wetted components nearby.

This preferred configuration permits the lander to support the heavy parts of the MAV during earth launch and Mars ascent. The thin tank wall only needs to support itself, perhaps aided by a small internal pressure. The tank's smooth upper hemisphere serves as an ascent fairing. A potential drawback of the parts arrangement is that the center of mass is aft of the center of pressure. However, destabilizing aerodynamic moments will be low on Mars.

Only an external tank pressurant tube could potentially interfere with the supersonic flow around the front of the vehicle. Instead of running it outside to the top of the tank, the tube may enter from below. Its length as shown in the sketch keeps bubbles away from the pump inlet.

The single tank MAV avoids large structural elements entirely. The only non-wetted flight structure is numerous brackets welded to the tank wall, as traditionally done on the sheet metal tank of the Atlas launch vehicle. For example, the threads visible in Figure 1 at the top of the thrust chamber would screw into titanium cylindrical shells welded directly to the tank.

Attitude Control

During ascent, 3-axis attitude control is effected by off-pulsing thrusters. The nozzles in the lower sketch of Figure 3 are appropriately labelled. Members of opposite pairs on two perpendicular axes would individually (depending on sign) control pitch and yaw. The remaining four thrusters would be angled slightly so that each opposite pair would produce a pure roll torque when shut

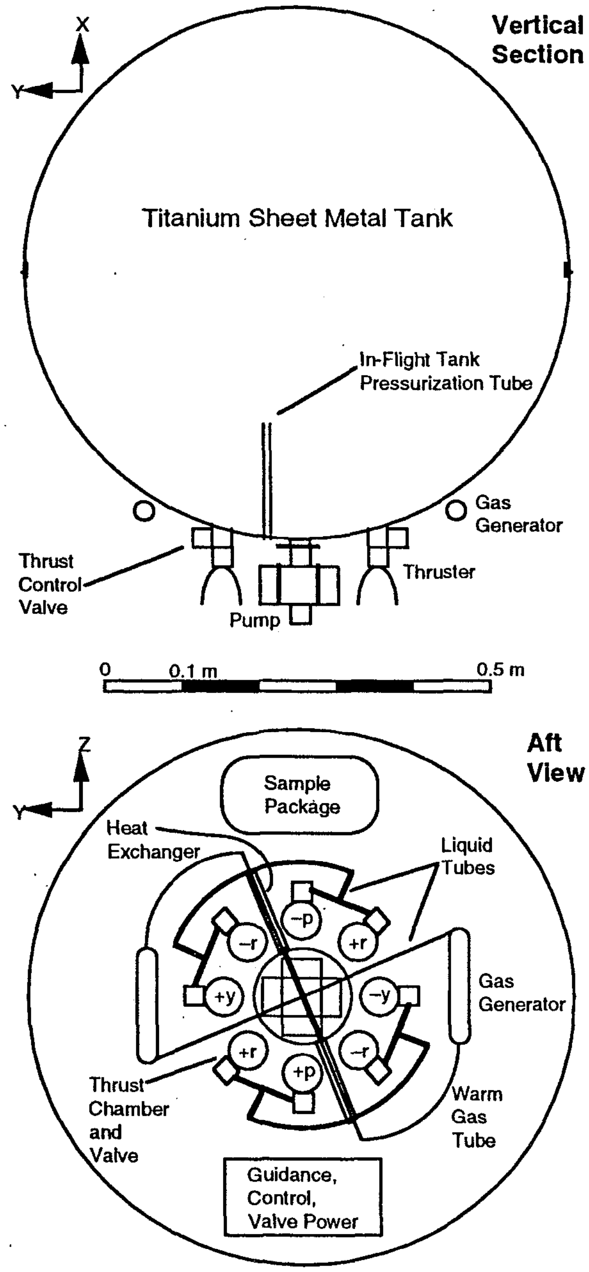


Figure 3. Locations of Major Components.

off together. Given eight thrusters, there is also the possibility of "engine-out" conrol capability with more sophisticated algorithms.

Torques of 25 N-m impart angular accelerations of 25 rad/s^2 to the ~1 kg-m^2 empty rotary inertia. Off pulsewidths of 5 msec therefore change the vehicle's angular velocity by 0.12 r/s. Ten guidance updates per second result in angular excursions on the order of .01 radian. The atmospheric part of the flight is smoother, as the propellant contributes some rotational inertia early on. The pitch and yaw thrusters could be angled toward an average c.g. location to reduce moment arms for an even smoother flight. On the roll axis, thruster mounting angles can be selected to obtain desired torques.

Tank

A conventional satellite tank sized for the MAV volume would consist of about 10 kg of alloyed titanium in a spherical shape. It would have an average wall thickness near 1 mm and a burst pressure above 3.5 MPa (500 psi). A reasonable goal here is to diminish the mass by a factor of 5, as a similar reduction in burst pressure is acceptable. Thus the tank wall material need not be stronger, just thinner. In the absence of earth safety concerns, reducing thickness is mainly a fabrication project. Options are to form hemispheres from thin sheet stock, or the traditional forge-and-machine approach. It is likely that either could be made to work. In the absence of meeting formal fracture mechanics requirements, the risk of an operational burst failure increases from negligible to acceptably low.

An inner diameter of 0.7 m (27.6 in) encloses over 179 liters. Considering hydrazine's thermal expansion, 175 kg of liquid will fit at temperatures up to 330 K (134 F). Worst case design calculations here assume that the tank wall has the density and low strength of pure titanium. The latter facilitates cold forming of hemispheres from sheet material. A wall thickness of 0.25 mm (.010 in) yields a shell mass of 1.75 kg and a hoop stress equal to 700 times the internal pressure. Thus at 0.35 MPa (50 psi) operating pressure, the tensile stress is very low at 245 MPa (36 ksi). Burst pressure is twice the operating pressure. This is a far greater margin than human-rated launch vehicle tanks, which are tested to only single-digit percentages above their similar operating pressures.

An equatorial ring would be used to facilitate welding the shell halves together. At a cross sectional area of 20 mm^2, its mass is 200 grams. To minimize residuals, 300 grams is budgeted for anti-slosh baffles near the outlet. The tank mass goal is therefore set at 2.3 kg, including a 50 gram porting allowance. Note that if the tank halves are machined from forgings in the traditional manner, wall thickness variations will influence achievable weight. The Ti-6Al-4V alloy is much stronger, so the minimum wall thickness in this case might be less than specified above.

Tank technology heritage, at least at the proof-of-principle level, comes from the pumped hydrazine flight in 1994. Titanium sheet 0.2 mm (.008 in) thick was welded with a Nd-YAG laser. Tanks were proof tested at a hoop stress above 620 MPa (90 ksi), and one was cycled hundreds of times to 525 MPa (76 ksi) without any failures except in deliberate burst tests. Non-recurring engineering (NRE) and special tooling are needed for the MAV tank, so it could cost more than a newly designed satellite tank. Finally, it should be noted that if there are insurmountable development or safety problems, doubling tank thickness would use only about half the mass margin in Table 4.

Quad Piston Pump

Figure 4 shows the quad piston pump assembly designed and built at LLNL in 1993. Four working cylinder assemblies are bolted to a central liquid manifold block which contains inlet and outlet check valves. This arrangement lowers liquid pressure losses as well as mass. Opposite pistons stroke toward each other, which cancels net mass shifts to greatly reduce vibration. There is no external control, as the gas valves are synchronized pneumatically. Piston speed and switching frequency can vary all the way down to zero, at full pressure. Actual flow depends entirely on thrust chamber valve actuation, just as in a pressure fed system.

Table 6 indicates sizing and performance for the hardware pictured. The 365 gram assembly delivered its own mass in liquid each second above 6.2 MPa (900 psi), from a tank at 0.35 MPa (50 psi). This flow of hydrazine would support vacuum thrust 230 times the pump's earth weight. Comparing the last lines of Tables 5 and 6 indicates that the pump cycle life requirement is immediately within reach.

For operation in a gas generator cycle engine, a key performance parameter is the pressure ratio of the liquid discharge to the driving gas. This boost ratio falls in the graph as flow rises. Boost is reduced by pressure losses in passageways during the power stroke, particularly the liquid discharge check valves and the gas intake. Even the static boost ratio (1.50) was below the piston area ratio (1.56), because gas leakage required intake flow.

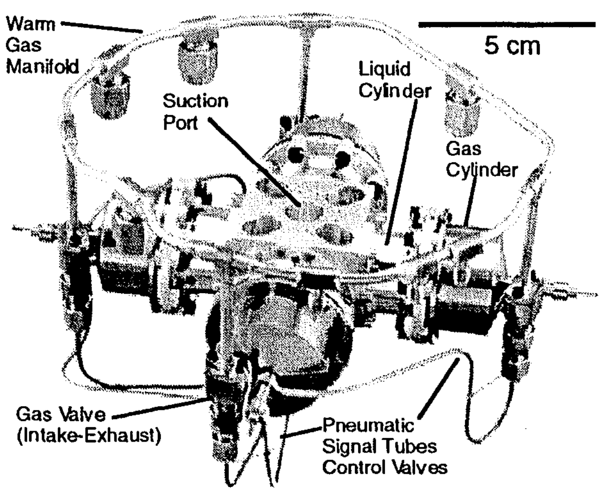


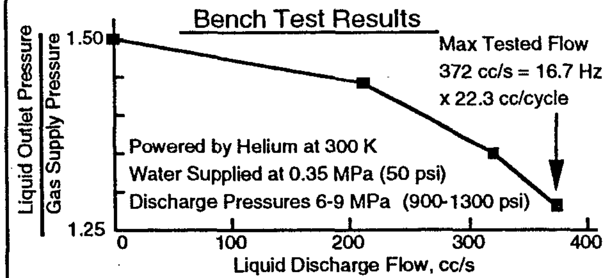
Figure 4. Four Chamber Hydrazine Pump Tested in 1993.

Table 6. Characteristics of the Quad Piston Pump.

Sizing Information

Mass As Tested
Liquid Cylinder Bore
Gas Cylinder Bore
Piston Area Ratio
Max Piston Travel
Ideal Liq Displacement

Bench Test Results



Tests with Hydrazine

Pump Fed Engine Static Test, 520 N Thrust at Sea Level
Pump Fed Engine Flight Test, 250-260 cc/s for 37 s Duration
Half Quad Life Test, Warm Gas & Water, >1500 cycles

The MAV pump would need to be rated at 3 to 4 times the flow of the original quad assembly. To hold pressure drops constant, areas must be scaled as flows to leave fluid velocities unchanged. Therefore, the linear dimensions of the pump could theoretically double.

Scaling mass as the cube of linear dimensions yields a 2 to 3 kg estimate. However, the piston travel would not need to double as passageway diameters do. Both mass and overall size would be reduced significantly by machining the liquid cylinders and block as one piece. Dimensions bounded by fabricability would not be thickened in the larger pump. For example, tubing walls in Figure 4 are many times too strong. Given these lightweighting opportunities, the MAV pump mass would be 2 kg or less.

The gas valves in Figure 4 vent directly to ambient with no exhaust manifold. Considering the central aft location of the MAV pump in Figure 3, it would be simple to run the exhaust through aftward-pointed nozzles with little extra mass. While this would contribute extra thrust, no credit was taken for it in the system analysis. Any exhaust flow restriction would have to be taken into account. Specifically, the relation between pump refill rate and tank pressure depends on the restrictiveness of the exhaust as well as the liquid inlet check valves.

A technical obstacle to perfecting a highly reliable system was the need for dynamic warm gas seals on the pump pistons and in the intake-exhaust valves. Solid graphite seals were used near 1000 K and simply allowed to leak even more gas than was needed for pump power. Reducing leakage will not only help consumption, it will shift the pressure boost curve upwards. Potential advances were identified such as improving cylinder roundness and stiffness, while relying more on metal seats than on sliding seals in the valves.

At the opposite extreme, fully cooling the gas to 300 K would guarantee a long lifetime for elastomer seals. The heat exchanger would be heavier, and system Isp would fall to 231 s. One kilogram of stage hardware would need to be replaced by propellant. This relatively small number shows it's not critical to seek an ideal optimum, so the compromise used in Table 5 is 600 K.

Concepts already being tested for liquid cooling of soft pump seals could permit elevated gas temperatures. In particular, a nontoxic propulsion system recently tested at LLNL uses a liquid-cooled pump driven by decomposed hydrogen peroxide. Reliable operation has been demonstrated on numerous occasions without gas leakage, and without refurbishment. The technology would be applicable to operation with hydrazine.

A new prototype quad pump being designed is intended to be simpler and cheaper than the hardware pictured in Figure 4. Also, it is operationally more robust with regard to the pneumatic logic for valve switching.

Gas Generator

Like the MR-125 thruster, this technology can be considered mature. A long catalyst bed is used to promote endothermic ammonia dissociation, thus cooling the gas and reducing average molecular weight. A Primex high pressure gas generator was developed specifically to power the LLNL reciprocating pump. At 115 grams, it includes a small integral accumulator and a filter. Its typical throughput is 5 g/s at 0.9 MPa pressure drop. One was tested to a total throughput of 0.95 kg. Four of them in parallel would meet both flow and life requirements in Table 5.

A pair of larger gas generators could be used as shown in Figure 3. Choosing either two or four is consistent with evenly feeding the pump gas manifold. Placement is not critical, so the g.g.'s could be moved for vehicle balancing.

Propulsion Mass and Heritage Summary

The masses of the remaining components have similarly been estimated based on functional parts tested previously. As listed in Table 7, they are below 100 grams each, so further details are omitted here for the sake of brevity. The table indicates that propulsion hardware can weigh even less than the goal in Table 4.

Table 7. Mass Summary for Fluid Parts and Structure.

Component		Mass, kg
Tank		2.3
Pump		2.0
Thrusters	(8 at 100 grams)	0.8
Thruster valves	(8 at 50 grams)	0.4
Gas Generators	(4x 115 g or 2x 230)	0.46
Gas Gen Valve		0.05
Liquid Regulator		0.05
Heat Exchangers	(2 at 50 grams)	0.1
Gas Regulator		0.05
Check Valve		0.05
Pressurant Tube		0.1
Flyaway Fluid Disconnect		0.1
Liquid Filters		0.2
Liquid Lines		0.04
Gas Lines		0.03
Fluid Joints		0.25
Thruster Mounts		0.12
Component Brackets		0.25
Propulsion Hardware Total		7.35 kg

There is a historical record for system operation as well as component masses. Table 8 summarizes what was achieved for under $20 million spent between 1988 and 1994. Unique new components were developed from innovative ideas, then pumped hydrazine systems were successfully operated on several occasions including the flight of a miniature vehicle. The results have been documented in detail.[5-7]

Table 8. Key Elements of Demonstrated Capability.

LLNL ASTRID Flight at Vandenberg AFB, 1994
• Miniature Pumped Hydrazine Launch from a Planetary Surface • Vacuum Δv Equivalent >2000 m/s for Just 21 kg Liftoff Mass • Isolation Valves Omitted for Less Mass and Complexity • Lightweight Quad Configuration for Piston Pump • Titanium Sheet Metal Tank, Wall Thickness 0.2 mm (.008 inch) • Warm Gas Pressurized Tank, No Gas-Liquid Separator • Thrusters Individually Pulsed for Warmup Prior to Liftoff • Soft Elastomer Seals Contained Warm Gas in Piloted Valves
ASTRID Static Fire Tests at VAFB, 1993
• Bootstrap Start from Only 0.22 MPa (32 psi) Tank Pressure • Thruster Off-Pulsing During Steady Burn, Re Attitude Control • Warm Gas Entering Thin Tank Reached 450 K without Incicent • Terrestrial Test Experience Relevant to MAV Development
Components and Systems at Primex Test Lab
• High Pressure Long Life Thruster Proven Repeatedly • Smooth Pump System Operation in Highly Instrumented Tests • Leaky Pump Seals Detracted Just 5% from System Isp • Thruster Pulses Down to 5 msec During Pump-Fed Operation
Laboratory Tests at LLNL
• High Power-to-Weight Measured for LLNL Developed Pumps • Thin Wall Tube Used for Mini Heat Exchanger at 7 MPa, 900 K • Leaktight Liquid Cooled Warm Gas Seals Work in H2O2 Pumps

Broader Aspects of MAV Design

The goal herein has been to treat one particular propulsion option in detail. The many related mission problems include sample handling mechanisms and biological contamination issues. Thus a complete MAV design is beyond the scope of one propulsion paper. Nevertheless, several broader issues are discussed below to help put the propulsion design in context.

Scaling to Fit Mission Needs

The achievable mass of guidance and control hardware is likely to determine a minimum MAV size. It would be straightforward to shrink the propulsion design to a Mars liftoff mass as low as 100 kg. Many components, such as the pump, would be sized closer to ones already tested. At 150 kg, six thrusters would be used, and four would lift a 100 kg MAV. Preserving the 1% tankage fraction at this latter size only requires thinning the wall to 0.2 mm (.008 in), which has been demonstrated.

Trajectory & Drag

It's necessary to determine a specific Δv requirement based on a detailed trajectory analysis to a particular orbital altitude. The trajectory may require attitude control during coasting and a restart capability for a circularization burn. In this case, microgravity fluid management and miniature gas jets would consume some of the mass margin. During any long coasts, the gas generator valve may be shut if there are leakage or heating concerns, or simply to conserve electrical energy. As long as low pressure liquid is available to the pump, full system pressure is rapidly obtained by simply reopening the gas generator valve.

Considering drag and stability during atmospheric flight, the supersonic flowfield around a sphere needs to be considered. Flow around the exposed components would remain subsonic. At Cd=1, a dynamic pressure near 250 N/m^2 (5 lb/ft^2) would produce 100 N of drag. This may occur for up to one minute, which would cancel the impulse delivered by 25 kg of hydrazine. The lost Δv was accounted for in Tables 3 and 4.

Alternative Vehicle Shapes and Staging

A single stage MAV having an ideal tank shape certainly simplifies the propulsion system. However, the indicated mass margin may allow for less efficient cylindrical tankage in a tall narrow MAV having one or more stages. These options would have to be examined in detail separately. Such a vehicle would require a tilt-up launch platform as suggested in Reference 3 and subsequently found to make sense for a multi-stage solid rocket.[4]

A key feature of solid-propelled MAV concepts is that the guidance package need not be carried to orbit. It may be worth applying this to liquid propulsion, e.g. by using pumped hydrazine for a spinning upper stage. As noted above, a lot depends on the mass of avionics.

Mars GSE

Ground support equipment must be included on the lander, just as is required for launch vehicles departing from earth. Removable structural, fluid, electrical, and thermal interfaces would conveniently all be underneath a spherical single stage MAV. It would essentially be nested within its support equipment, including heaters such as RHU's.

The electrical interface would load guidance information, and supply power until the moment of launch. As this would include solenoid power, valve operation could even be verified acoustically before fueling, without draining flight batteries. Solenoids could even be used as heaters.

If the MAV is not leaktight, the support equipment may include a regulated helium supply to maintain positive pressure during transport to Mars. Before fueling, the MAV must be vented. The hydrazine would be supplied from conventional tanks having heaters and insulation. Little of their volume would be wasted since a blowdown ratio as high as 10 could fill the MAV to its low 0.35 MPa starting pressure. A pressure transducer on the ground side is sufficient to indicate when propellant transfer is complete. Depending on the results of thermal analysis, and on the countdown timeline, the fuel might be heated above 320 K to preclude the possibility of prelaunch freezing (which occurs at 275 K). Even if frost were to form inside the unheated upper tank half, it would be melted by warm gas pressurization and aerodynamic heating.

No initial gas bubble is needed in the MAV, since the pump will bootstrap itself up while connected to the pressurized liquid source. In practice, residual gas at Mars ambient pressure would compress to well under a liter, leaving the flight tank over 99.5% full. Maintaining this level up until the moment of launch would occur without controls. Thus, warmup propellant for the gas generator circuit and thrust chambers is supplied by the ground tanks. As the MAV flies away, the withdrawal of the fueling tube lets a check valve close on the flight side.

Testing on Earth

A primary issue for earth testing is the potential exposure of people to toxic propellant contained inside very lightweight hardware. The risk of a significant explosion is minute, due to the small size and low tank pressure. One serious safety concern is that of flowing high temperature gas close to liquid hydrazine in the heat exchanger. This is a situation in which the physical likelihood of a problem can be shown to be low, but the consequences are nevertheless high. A remote test area would most likely be required for operating flightweight hardware.

Explosions do occur when heat is continually added to a fixed amount of hydrazine, e.g. in a tube with no flow. Under normal operation of the pump system, gas and liquid flow start and stop together. It would be straightforward to automatically shut the gas generator feed valve if a gas leak creates a hazard when hydrazine stops flowing. A heavyweight tank and water as coolant could easily be substituted during initial laboratory testing. The heat exchangers would ultimately be tested alone with hydrazine.

While most may be unaffordable, there are numerous potential flight test scenarios for a MAV and its support equipment. These include a suborbital ground launch, a suborbital high altitude flight (e.g. from a balloon), a flight to orbit from a booster or RLV prototype, and in space. Some of these suggest useful mission possibilities, such as GTO to the lunar surface with an increased payload.

Bipropellant and Solid Options Revisited

Among rocket professionals and propulsion users alike, it is natural to assume that higher Isp will readily increase the payload fraction for a given maneuvering requirement. The dark shaded bars in Figure 5 show the single stage non-expended mass allowances for three propellant options. By comparison, the improved performance of bipropellants increases the burnout mass of a 200 kg loaded MAV by 18 kg. This would indeed suggest a great increase in payload capacity if propulsion realities are not considered.

For each propellant, the known masses of associated hardware elements are displayed. The leftmost example is the single stage pumped hydrazine MAV. The mass breakdown shown next to the center bar is typical of conventional bipropellant technology flown on satellites. Clearly, the masses of propulsion hardware and structure can easily negate the theoretical advantage of increased Isp for a small-scale MAV. Also, bipropellants would most likely have a greater residual fraction as shown, since there must be a planned reserve in case mixture ratio varies.

The references describe both pump fed and pressure fed bipropellant technology options for trimming mass. However, all the lightweight bipropellant concepts advanced to date are sufficiently speculative that no mass numbers traceable to tested propulsion systems were available to include in Figure 5.

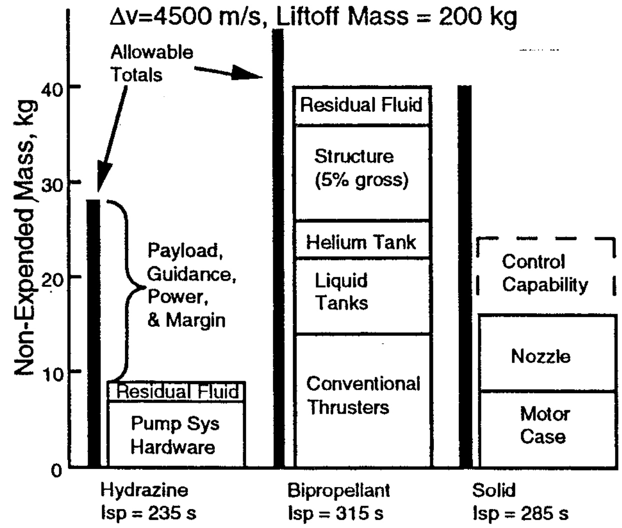


Figure 5. The Value of Isp for Single Stage Mars Ascent.

Numerous miniature high pressure bipropellant thrusters even smaller than the MR-125 have been demonstrated. Thrust-to-weight ratios have been exceedingly high, so it would be an obvious match to feed them with reciprocating pumps like the ones developed for hydrazine. Unfortunately, lightweight biprop thrusters that can run continuously for many minutes have not emerged from these developments. Even if appropriate thrusters existed, pumped hydrazine would still be a more mature technology than miniature pump-fed bipropellants.

Figure 5 includes the solid propellant option for completeness. Since multiple burns are needed to achieve orbit, this single stage comparison would represent a solid motor having restart capability. Accordingly, multiple grains and igniters would be needed. The graph indicates that the trade between solid propellant and hydrazine depends on control system mass.

The bar chart may be viewed as merely indicative of relative stage inert fractions for different propellants. Solid rocket motors consist of titanium spherical shells built to withstand operating pressures near 7 MPa (1000 psi). They also have high thrust levels due to inherently fast burn rates. Therefore, the motors tend to have heavy cases and large nozzles which each mass approximately 5% of the propellant. Fundamentally, it requires less material to build low pressure liquid tanks and engines sized for the maneuver.

The main attraction of solid MAV concepts is that new propulsion technology development is not necessarily required. Instead, there are vehicle development challenges such as spin dynamics issues and the need for lightweight auxilliary hardware to produce high control moments. The control mass uncertainty depicted in Figure 5 reflects this complexity. Additionally, staging may penalize solids more than liquids, due to the extra interstage structural mass required to surround large nozzles. Finally, it is noteworthy that solid propellant cannot be transferred on Mars. The structural hardware, and also the solid propellant grains themselves, must withstand the extreme acceleration environments.

As a final point for comparison, the engine exhaust of a hydrazine MAV will do the least damage to the Mars lander and its associated scientific instruments. There are lower temperatures, no organics, and no solid condensibles. Mars landers use hydrazine partly for this reason, and it makes sense for the MAV so that useful lander science can continue after launch.

Discussion

Affordable Mars Sample Return is a fascinating problem which requires challenging advances in miniature rocketry. A reliable MAV is going to be a unique new item. It would not be surprising if successful MAV implementation requires more time and funding than familiar mission elements which rely on mature technology bases. Mars landers have been implemented over periods spanning decades of time. Rover capabilities have also benefitted from long term development. If the need for propulsion innovation could be widely acknowledged, then it would make sense to test multiple options for the MAV in a timely manner.

Pumped hydrazine could be the most practical MAV propulsion scheme. This was considered several years ago, but it was necessary to study the bipropellant options first. In the interim, the answers to "why not biprops" have been documented in the specific context of attempted MAV designs which were not well received. Another recent event in mission planning was payload downsizing. This apparently occurred as a result of the LLNL-JPL joint study summarized by Reference 3. The new minimum scale philosophy should enhance the acceptability of a small monopropellant MAV.

There are many applications for flexible liquid propulsion technology which can lift rocks off Mars. They include both landing on and launching from earth's moon. There is also relevance to planetary micromissions which need significant maneuvering for tiny spacecraft starting from GTO. Multiple uses present the opportunity to share development costs broadly across exploration budgets. The pumped hydrazine technology in particular is an obvious way to lighten Mars landing propulsion systems. This suggests another possibility for future study. Given more propellant in larger tanks, a Mars landing propulsion system might be useful as a first ascent stage.

Conclusion

Feeding proven high pressure thrusters from a thin tank using a pump enables a single hydrazine stage to lift Mars samples to orbit. In the opinion of the authors, this option offers an acceptably low risk, affordable, and technically defensible approach to Mars Sample Return. This claim is strongly supported by a detailed MAV design concept which relies upon previously documented technology development efforts.

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